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FEASIBILITY OF A SUSTAINER PROJECTILE IN THE 30-MM, 35-MM, AND 40-MM CALIBER RANGE

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A sustainer projectile is a projectile which sustains its initial launch velocity while in flight by the use of rocket propulsion to overcome drag. The nomenclature for this type of system in the larger calibers is RAP (rocket-assisted projectile). This report, then, does not deal with a new concept but addresses the feasibility of RAP's in the smaller calibers appropriate for air defense, i.e., the 30-40 mm caliber range. The nominal objective of this study was to examine the feasibility of sustainer technology reducing the time of flight to 3 s or less at 3 km. This could be achieved by a 35-mm candidate and a 40-mm

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20. Abstract (contd):

candidate; however, the 30 mm is too small to support the necessary propulsion system.

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I. INTRODUCTION

A sustainer projectile is a projectile which sustains its initial launch velocity while in flight by the use of rocket propulsion to overcome drag. This concept is not new in ballistics, for some large caliber artillery rounds employ such a system and are aptly called rocket-assisted projectiles (RAP's). This report, then, does not deal with a new concept, but addresses the feasibility of RAP's in the smaller calibers appropriate for air defense. Sustaining the muzzle velocity has the advantage of decreasing the projectile time of flight, thus reducing the time available to the target aircraft for maneuver. If the aircraft has less time to maneuver, the probability of the projectile hitting it is increased, since the aircraft will be closer to its predicted position. The nominal objective of this study was to examine the feasibility of sustainer technology reducing the time of flight to 3 s or less at 3 km. The 30-mm Minneapolis-Honeywell GAU-8, the 35-mm Oerlikon and the 40-mm Bofors are used as the reference guns/projectiles. The sustainer design is to change the reference projectile as little as possible. Internally, the rocket propellant replaces some of the high explosive (HE) in the cavity. The necessary throat nozzle assembly must be fitted into the remaining volume. Externally, the total round mass, before burning, must be equal to the reference mass. In addition, the shape of the round must remain the same. An assumption is made that the new projectile has the same drag coefficient as the reference projectile.

II. DETERMINATION OF ROCKET PROPELLANT MASS

We determine the flight time to a given range, R, as follows:

$$t_f = t_b + t_{nb} \tag{1}$$

where t_f = time of flight
t_b = time while burning
t_{nb} = time while not burning

$$R = R_b + R_{nb} \tag{2}$$

where R = total range
R_b = range while burning
R_{nb} = range while not burning.

then,

$$t_b = \frac{R_b}{v_m} \tag{3}$$

where v_m is the muzzle velocity.

To determine the time spent in free flight, we adopt the Siacci approximation:

$$t_{nb} = \frac{R - R_b}{v_m - c\,(R - R_b)} \,. \tag{4}$$

In the calculations we have assumed that the Siacci drag-related coefficient, c, is the same as for the reference projectile.

Now, the drag force to be overcome by the rocket propellant is given

$$D = \frac{1}{2} C_D A \rho_a v_m^2 \tag{5}$$

with C_D = drag coefficient

A = cross-sectional area of projectile

ρ_a = density of air.

The burn time is dependent on the exit velocity of the gases, the drag force and mass of the propellant

$$t_b = \frac{v_e m_p}{D} \tag{6}$$

where v_e = exit velocity of gases

m_p = propellant mass

and

$$v_e = I_s g \tag{7}$$

where I_s = specific impulse of propellant

g = gravitational acceleration.

By combining these equations we obtain explicit equations for the time of flight to range R in terms of m_p and v_m:

$$t_f = \frac{R}{v_m}, \quad R < v_m t_b$$

$$t_f = \frac{\alpha m_p}{v_m^2} + \frac{R - \alpha \frac{m_p}{v_m}}{v_m - \beta (R - \frac{\alpha m_p}{v_m})}, \quad R > v_m t_b \tag{8}$$

where for convenience we have used the notation

$$\alpha = \frac{I_s g}{\frac{1}{2} C_D A \rho_a} .$$

The numerical values to be inserted into Equations (8) are shown in Table 1.

The times of flight to R=3 km for the three reference-caliber projectiles are plotted in Figure 1 as a function of m_p. The time of flight to 3 km for each of the conventional projectiles is shown by the horizontal lines. We observe that about 90 gm of propellant is required in the 40-mm projectile to achieve our objective of no greater than 3 s time of flight. For the 35-mm projectile we can apparently exceed the nominal objective with 60 gm of propellant. For the 30-mm projectile we require 20-25 gm of propellant.

We now turn to the feasibility of containing the required propellant mass in the available volume, burning it properly to achieve the desired propulsion, and designing a rocket assembly capable of fitting into the base of the projectile.

III. ANALYSIS AND ROUND DESIGN

The configurations of the rocket propellant grain and the rocket assembly were determined from the sets of equations described below. These equations along with reasonable input values for the solid propellants and nozzle assemblies were obtained from Reference 1. The interaction of the results determines the exact shape and length needed.

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[1] George P. Sutton and Donald M. Ross, Rocket Propulsion Elements, 4th Edition, John Wiley & Sons, 1976.

TABLE 1. INPUTS FOR TIME OF FLIGHT TO RANGE, R, CALCULATIONS

Constants

$g = 9.8\ (m/s^2)$

$\rho_a = 1.225\ (kg/m^3)$

$I_S = 260.0\ (s)$

30 mm

$v_m = 1025.0\ (m/s)$

$\beta = 0.094\ (s^{-1})$

$C_D = 0.047$

35 mm

$v_m = 1175.0\ (m/s)$

$\beta = 0.129\ (s^{-1})$

$C_D = 0.1295$

40 mm

$v_m = 1035.0\ (m/s)$

$\beta = 0.105\ (s^{-1})$

$C_D = 0.158$

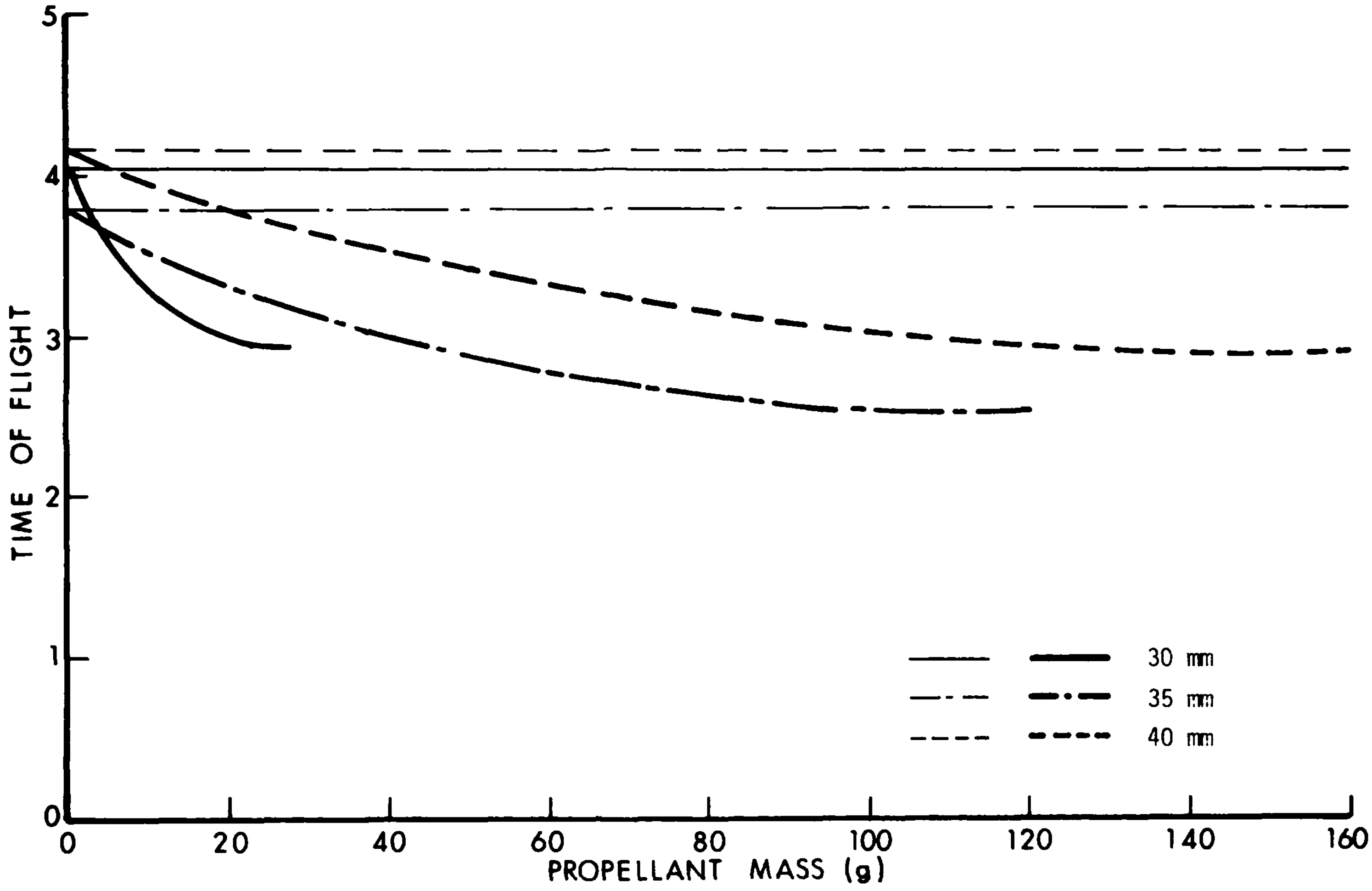

Figure 1. Time of Flight to 3 km vs Propellant Mass

The volume required for the propellant is determined from the mass and density of the rocket propellant,

$$V_p = m_p/\rho_p \tag{9}$$

The web thickness, representing the minimum required thickness of propellant from the initial burning surface to the insulated case wall, is defined as the burn time multiplying the burning rate of the propellant,

$$w = t_b \, r_b \tag{10}$$

Since an end-burning grain (cigarette-style) is considered an inefficient configuration due to its axial-only burning which requires longer grains, a hollow cylinder configuration was chosen. Twice the web thickness is subtracted from the outside diameter, to obtain the diameter of the hollow cylinder in the grain

$$d_i = d_o - 2w \quad . \tag{11}$$

The length of the grain is then computed by dividing the volume of the rocket propellant by the area of the annulus

$$\ell_g = \frac{V_p}{\frac{\pi}{4}\left(d_o^2 - d_i^2\right)} \quad . \tag{12}$$

The ℓ/d ratio is then just

$$\frac{\ell}{d} = \frac{\ell_g}{d_o} \; . \tag{13}$$

The web fraction helps to determine if fancy patterns are needed in the grain, such as a star or wagon wheel, to achieve proper burning. The equation

$$w_f = 2w/d_o \tag{14}$$

yields the resultant fraction. The average burn area is computed from the mass and density of the propellant and the burn time and rate:

$$A_b = \frac{m_p}{t_b \, \rho_p \, r_b} \; . \tag{15}$$

The next set of equations determines the parameters for the nozzle. The area of the throat is computed from

$$A_t = \frac{\rho_p A_b r_b I_S}{P C_F c_F} \qquad (16)$$

where P = pressure at nozzle, taken nominally equivalent to 1000 psi

C_F = optimum thrust coefficient

c_F = thrust correction factor.

From this result the diameter of the throat is obtained where A_t is taken as a circular cross section, i.e.,

$$d_t = \sqrt{4A_t/\pi} \ . \qquad (17)$$

The exit area is calculated by multiplying the throat area by a factor epsilon, determined from various pressure and specific heat ratios, resulting in

$$A_e = A_t \varepsilon \ . \qquad (18)$$

The exit diameter, d_e, can be calculated using Equation (17) by substituting A_e for A_t. The nozzle length from throat to exit is calculated using a constant, specific for the A_e/A_t ratio just calculated and a contour nozzle, and the exit diameter yielding

$$\ell_n = 2.8\ d_e \ . \qquad (19)$$

These equations were evaluated for the required m_p's and for a variety of solid propellants. It was found that the 30-mm projectile could not provide adequate volumes and lengths to accommodate a sustainer system with any of the operational propellants considered. Thus all attention was focused on the 35-mm and 40-mm projectiles. One round/propellant combination for each caliber was obtained which met all constraints. These configurations are displayed in Figures 2 and 3. The specific parameters are listed in Tables 2 and 3. No calculations were done on the gyroscopic stability of these rounds; therefore they could be unstable in flight.

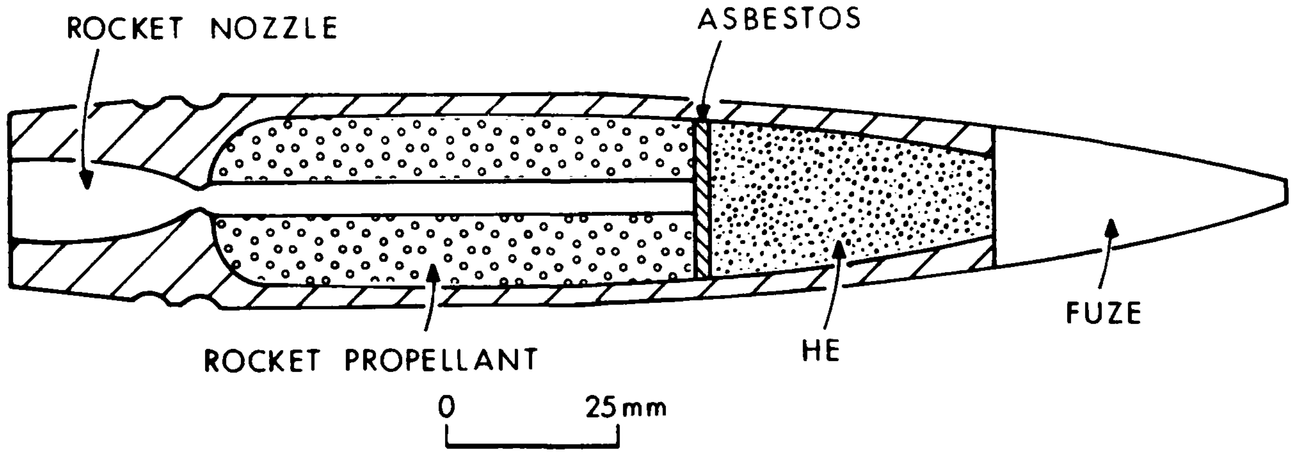

Figure 2. Proposed 35-mm Sustainer Projectile

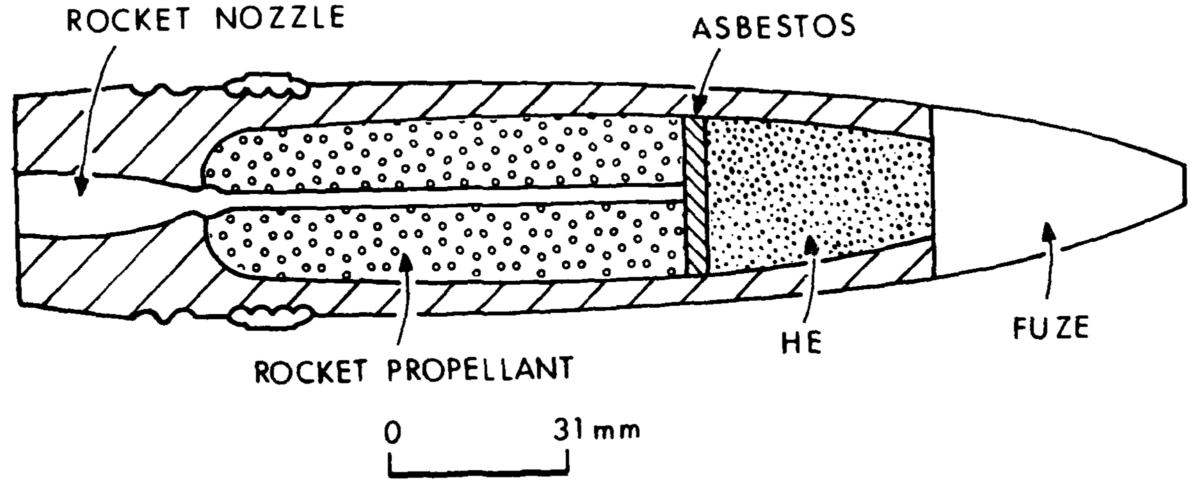

Figure 3. Proposed 40-mm Sustainer Projectile

TABLE 2. PARAMETERS FOR THE PROPOSED 35-MM SUSTAINER PROJECTILE

Trajectory Variables	
m_p (kg)	0.060
t_b (s)	1.45
R_b (m)	1704.9
Propellant Characteristics	
Composition	Polyurethane/Ammonium Perchlorate/Aluminum
I_S (s)	260.0
ρ_p (kg/cm^3)	0.00177
r_b (cm/s)	0.6858
Grain Configuration	(Internal Burning Conocyl)
V_p (cm^3)	33.83
w (cm)	1.00
d_o (cm)	2.40
d_i (cm)	0.41
ℓ_g (cm)	7.70
ℓ/d	3.21
w_F	0.83
A_b (cm^2)	34.00
P (dyn/cm^2)	6.891×10^7
C_F	1.57
c_F	0.92
A_t (cm^2)	0.1057
d_t (cm)	0.367
A_e (cm^2)	0.846
d_e (cm)	1.038
ℓ_n (cm)	2.90

TABLE 3. PARAMETERS FOR THE PROPOSED 40-MM SUSTAINER PROJECTILE

Trajectory Variables	
m_p (kg)	90.0
t_b (s)	1.76
R_b	1821.6

Propellant Characteristics

Composition Polyurethane/Ammonium Perchlorate/Aluminum

I_S (s)	260.0
ρ_p (kg/cm^3)	0.00177
r_b (cm/s)	0.6858

Grain Configuration (Internal Burning Conocyl)

V_p (cm^3)	50.75
w (cm)	1.21
d_o (cm)	2.70
d_i (cm)	0.29
ℓ_g (cm)	8.96
ℓ/d	3.32
w_F	0.89
A_b (cm^2)	42.04
P (dyn/cm^2)	6.891×10^7
C_F	1.57
c_F	0.92
A_t (cm^2)	0.131
d_t (cm)	0.408
A_e (cm^2)	1.048
d_e (cm)	1.155
ℓ_n (cm)	3.23

LIST OF SYMBOLS

A	cross-sectional area of projectile, m^2
A_b	solid propellant burning area, cm^2
A_e	nozzle exit cross-sectional area, cm^2
A_t	nozzle throat cross-sectional area, cm^2
C_D	drag coefficient
C_F	optimum thrust coefficient
c_F	thrust correction factor
D	drag force, N
d_e	nozzle exit area diameter, cm
d_o	propellant grain outer diameter, cm
d_i	propellant grain inner diameter, cm
d_t	nozzle throat diameter, cm
g	gravitational acceleration, m/s^2
I_S	specific impulse, s
ℓ_g	propellant grain length, cm
ℓ_n	nozzle length, cm
ℓ/d	propellant grain length to diameter ratio
m_p	propellant mass, kg
P	pressure at nozzle, cyn/cm^2
R	range, m
R_b	range while burning, m
R_{nb}	range while not burning, m
r_b	propellant grain burn rate, cm/s
t_b	propellant grain burn time, s
t_f	total flight time, s
t_{nb}	propellant grain nonburn time, s
v_e	nozzle exit velocity, m/s
v_m	muzzle velocity, m/s
V_p	propellant volume, cm^3
w	web, cm
w_F	web fraction

LIST OF SYMBOLS (contd)

	convenient fractional representation
	Siacci drag-related coefficient
	nozzle exit to nozzle throat area ratio
a	air density, kg/m^3
p	propellant density, kg/cm^3

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